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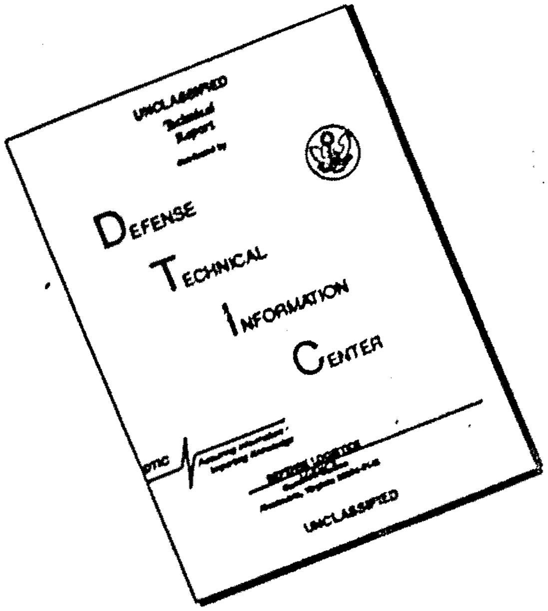

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USAAVLABS TECHNICAL REPORT 69-10B

ADVANCEMENT OF SMALL GAS TURBINE COMPONENT TECHNOLOGY (U)

ADVANCED SMALL AXIAL COMPRESSOR (U)

VOLUME II - ADDENDUM
AERODYNAMIC REDESIGN (U)

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By

James V. Davis

Edmund J. Dellert

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February 1970

U. S. ARMY AVIATION MATERIEL LABORATORIES
FORT EUSTIS, VIRGINIA

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CONTRACT DA 44-177-AMC-296(T)
CONTINENTAL AVIATION AND ENGINEERING CORPORATION
DETROIT, MICHIGAN

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(U) The research described herein, which was conducted by Continental Aviation and Engineering Corporation, was performed under U.S. Army Contract DA 44-177-AMC-296(T). The work was performed under the technical management of Mr. David B. Cale, Propulsion Division, U.S. Army Aviation Materiel Laboratories.

(U) This document is the classified addendum to USAAVLABS Technical Report 69-10B.

Task 1G162203D14413
Contract DA 44-177-AMC-296(T)
USAAVLABS Technical Report 69-10B
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ADVANCEMENT OF SMALL GAS TURBINE COMPONENT TECHNOLOGY (U)

ADVANCED SMALL AXIAL COMPRESSOR (U)

VOLUME II - ADDENDUM
AERODYNAMIC REDESIGN (U)

Continental Report 1033

By

James V. Davis
Edmund J. Dellert

Prepared By

Continental Aviation and Engineering Corporation
Detroit, Michigan

for

U. S. ARMY AVIATION MATERIEL LABORATORIES
Fort Eustis, Virginia

(U) SUMMARY

The two-stage axial compressor was redesigned using test data from the previous two stage tests and from an analysis of a family of high-pressure-ratio axial compressor rotors. The major changes made to the original design were reduced solidity in both rotors, increased first-stage rotor aspect ratio, and modified first-stage rotor blade profile.

The redesigned axial compressor had the following design point performance goals:

- Airflow = 5 lbs/sec
- Pressure ratio = 3.0:1
- Adiabatic efficiency = 82.6 percent
- Inlet hub tip ratio = 0.51
- First-stage rotor tip speed = 1448 ft/sec

(U) FOREWORD

This report presents the redesign analysis of the axial compressor program for the advancement of small gas turbine component technology. The redesign analysis is presented as an addendum to Volume II.

This program was sponsored by the United States Army Aviation Materiel Laboratories under Contract DA 44-177-AMC-296(T), Task 1G162203D14413.

The compressor program was divided into three phases. Phase I presents a study of a family of advanced axial compressors. It is reported as Volume I, with an addendum under a separate cover reporting the analysis and design.

Phases II and III are presented in Volume II. Phase II presents the axial compressor fabrication and test. Phase III presents the axial compressor redesign, fabrication and test.

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(C) DETAILED AERODYNAMIC REDESIGN (U)

(U) The compressor rotor under study in this program is shown in Figure 1. The redesign of the compressor rotor, first-stage was initiated based on the analysis of the family of high-pressure-ratio axial compressor rotors using the following ground rules:

Parameter	Range
Tip Solidity	1.3 to 1.45
Aspect Ratio	Greater than 0.8
Free Vortex Diffusion Factor	Less than 0.5

(U) Five redesign flowpaths were generated. Each flowpath was constructed to cover the range of hub and tip contours capable of matching with the existing second-stage flowpath contour. The aerodynamic parameters resulting from computer runs for each flowpath were analyzed and compared to provide direction to the final redesign flowpath. The optimum flowpath, Figure 2, was selected on the basis of rotor tip relative Mach number, tip turning, tip loading, and stator hub absolute Mach number. The rotor losses used in this study were obtained from the high-pressure-ratio axial compressor analysis.

(U) Using the optimum flowpath, the first-stage rotor exit pressure profile was varied to determine the effect of this parameter on the second-stage rotor incidence angles and the compressor exit velocity gradient. From this analysis, it was determined that a flat first-stage rotor exit stagnation pressure profile provided the best second-stage incidence match and compressor exit velocity profile.

(U) Throughout the redesign analysis, the Continental axial compressor aerodynamic design computer program was used to generate the static pressures and the corresponding velocities. This program included the effects of streamline curvatures and calculated the boundary layer growths based on flat-plate theory.

(U) Table I compares the redesign compressor performance with the original design values. A 1-percent transition duct loss, which was verified in Phase I testing, was used to determine the overall performance at the end of the transition duct. The differences in first-stage and overall efficiency between the original design and the redesign are directly related to the reevaluation of first-stage losses. In addition to redesign of the first stage, the aerodynamic implications of second-stage solidity were investigated in great depth. Significant

Figure 1. (U) Axial Compressor Rotor.

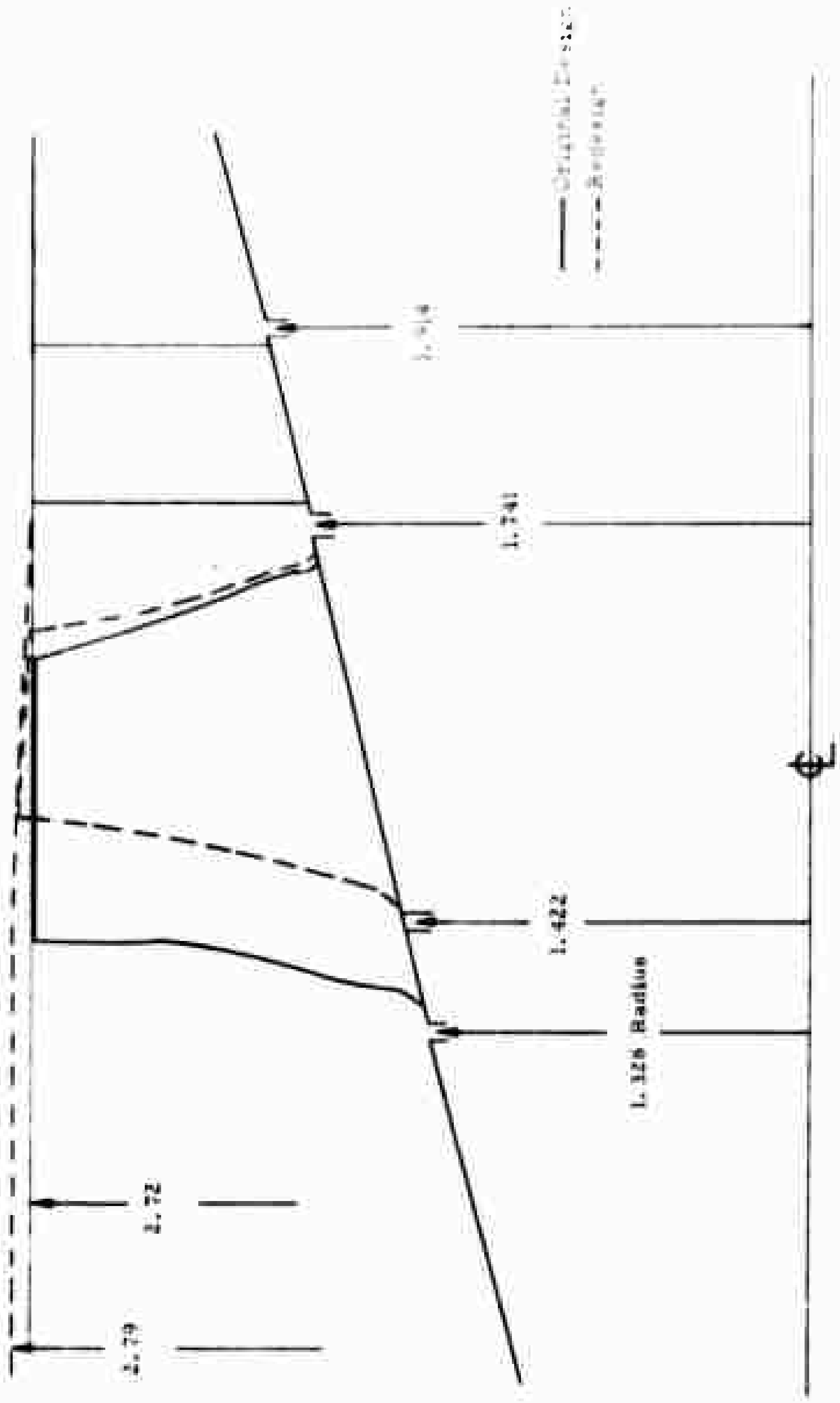

Figure 2. (U) Comparison of USAAVLABS F[illegible]-Stage Flow [illegible]

3

(C) TABLE I (U)

REDESIGNED COMPRESSOR PERFORMANCE COMPARISON (U)

	Original Design	First Stage Redesign	First Stage Redesign With Lower Solidity Second Stage
Compressor Rotational Speed, rpm	59,600	59,600	59,600
Compressor Flow Rate, lb/sec	5.0	5.0	5.0
Overall pressure ratio	3.0:1	3.0:1	3.0:1
Overall efficiency, %	82.3	81.1	82.6
Pressure Ratio - first stage	1.825:1	1.825:1	1.825:1
Pressure Ratio - second stage	1.660:1	1.660:1	1.660:1
Efficiency - first stage, %	85.5	83.9	83.9
Efficiency - second stage, %	82.9	82.9	86.1
Tip Speed - first rotor, ft/sec	1415	1451	1451

Performance at transition duct exit

performance increases were anticipated with a reduced solidity second stage (Table I) due to reductions in loss coefficient and increases in efficient flow range.

(U) The second-stage solidity analysis was conducted using the Continental double circular arc blade loss system. This loss system, which was recently placed in operation, predicts losses based mainly on diffusion and critical Mach number criteria. This method has successfully predicted performance of other compressors in the Mach number and diffusion range of the USAAVLABS second stage.

(U) As a consequence of the second stage solidity analysis, the number of blades in the second stage was reduced from 41 to 33, a solidity reduction of 19.5 percent. The blade contour of the original second stage was maintained.

(U) The losses used for the redesigned first-stage rotor, Figure 3, as compared to the original values, show a significant increase at the tip of the rotor and a slight decrease from the hub to about 80-percent span. This trend was observed in the test data for most of the rotors investigated in the high-pressure-ratio axial compressor study.

(C) The loss variation along the blade height significantly changes the local tip loading, as shown in a comparison with diffusion factors and relative velocity ratio, Figures 4 and 5, respectively. The value of tip diffusion factor for the redesigned rotor (0.56) appears to be high but has been demonstrated and exceeded on previous Continental high-pressure ratio rotors. For example, the first-stage rotor of Continental's 8:1 pressure ratio compressor has demonstrated diffusion factors in excess of 0.65 at comparable tip pressure ratio levels.

(C) The free vortex diffusion factor (the diffusion factor resulting from an assumption of constant spanwise pressure ratio and constant spanwise energy addition) is 0.423 as compared to 0.39 for the original design.

(C) Because of the increased redesign tip loss, the tip absolute flow angle into the stator increases from the original design value of 42 degrees to about 50 degrees (Figure 6). To match the tip of the stator to this higher angle, the stator was closed 8.5 degrees at the tip to zero degrees at a 2.53-inch radius (80-percent length).

(U) The redesigned first-stage pressure ratio and efficiency profiles are presented in Figure 7. The reduced efficiency at the tip is a result of the additional tip losses shown in Figure 3. The slight variation in stage pressure ratio is due to a reevaluation of first-stage stator losses (Figure 8), which was necessary because of the change in stator absolute angles.

(U) A comparison of the redesigned and original design exit velocity gradient, Figure 9, shows only minor change. The redesigned gradient should provide a satisfactory transition duct exit velocity profile.

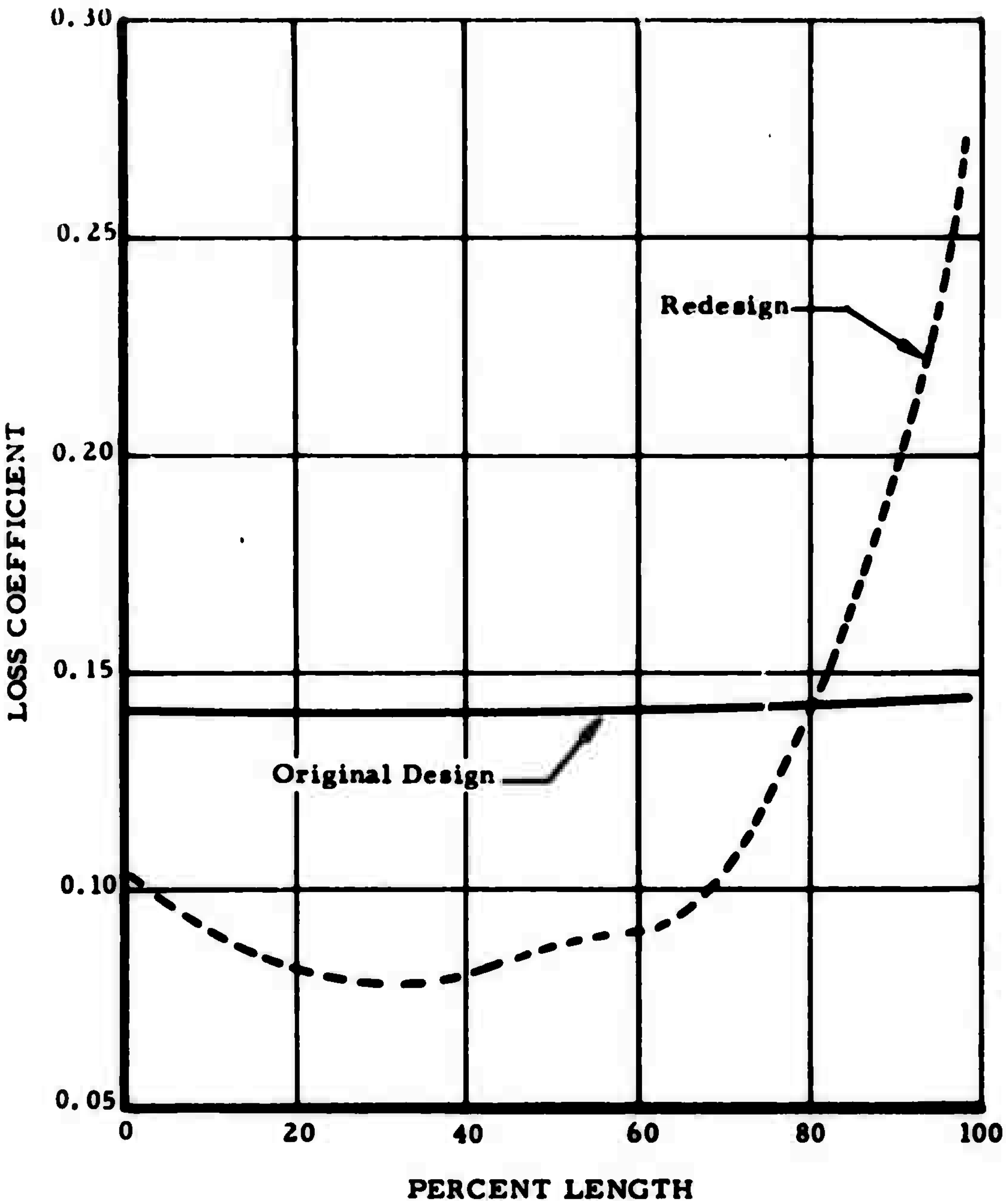

Figure 3. (U) First-Stage Rotor, Loss Coefficient Comparison.

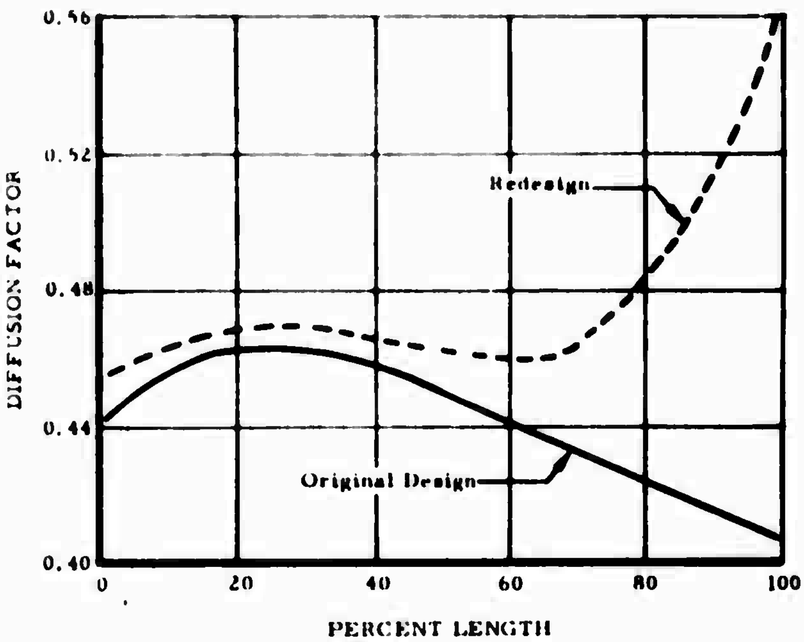

Figure 4. (U) First-Stage Rotor, Diffusion Factor Comparison.

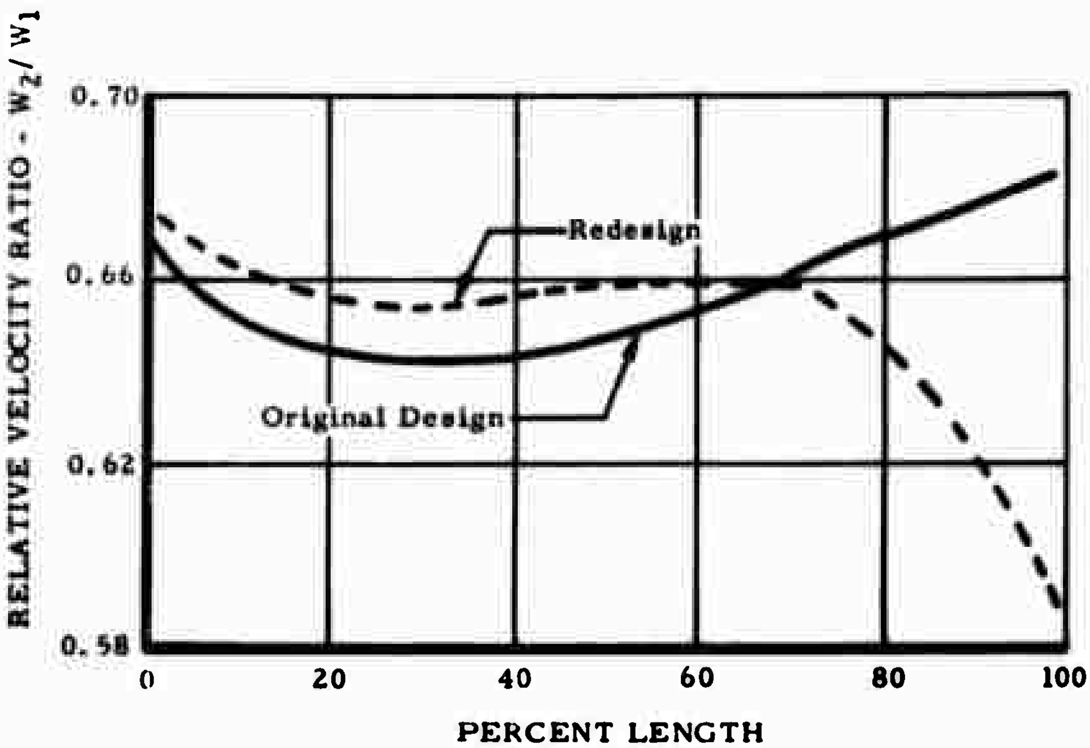

Figure 5. (U) First-Stage Rotor, Relative Velocity Comparison.

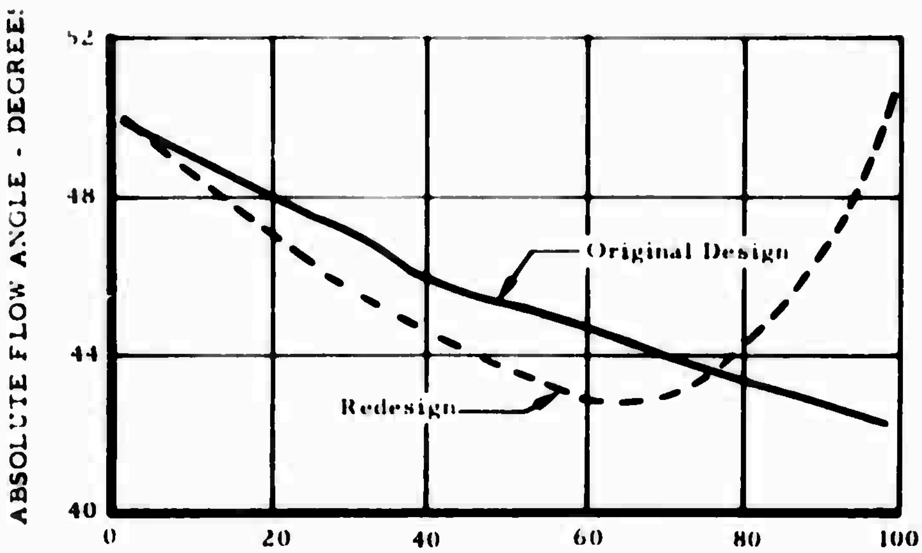

Figure 6. (U) First-Stage Rotor, Absolute Flow Angle Comparison.

(C) The redesigned flowpath, Figure 2, embodies a higher hub and higher tip radius at the inlet to the first stage. No change in flowpath contour was made downstream of the second rotor inlet. Increasing the aspect ratio from 0.646 to 0.963 required a higher first-stage inlet tip radius to provide a smooth flowpath transition between the first and second stage and to hold the tip relative inlet air angle between 65 and 70 degrees. Higher angles tend to close the rotor and narrow the rotor choke margin (the ratio of actual blade throat to the flow limiting throat, that is, sonic conditions), so that excessive incidence angles are required to pass the desired flow. Lower angles tend to raise the tip relative Mach number and in turn increase tip losses.

(U) As shown in Figure 2, an accelerating tip flowpath turn between the first-stage rotor and stator has been provided to reduce the tip loading. The accelerating turn tends to locally increase meridional velocities and to offset some of the velocity deceleration caused by high tip losses.

(U) The flow blockage factors (aerodynamic flow area divided by actual area) calculated for the redesign are listed in Table II and are compared to the original design values.

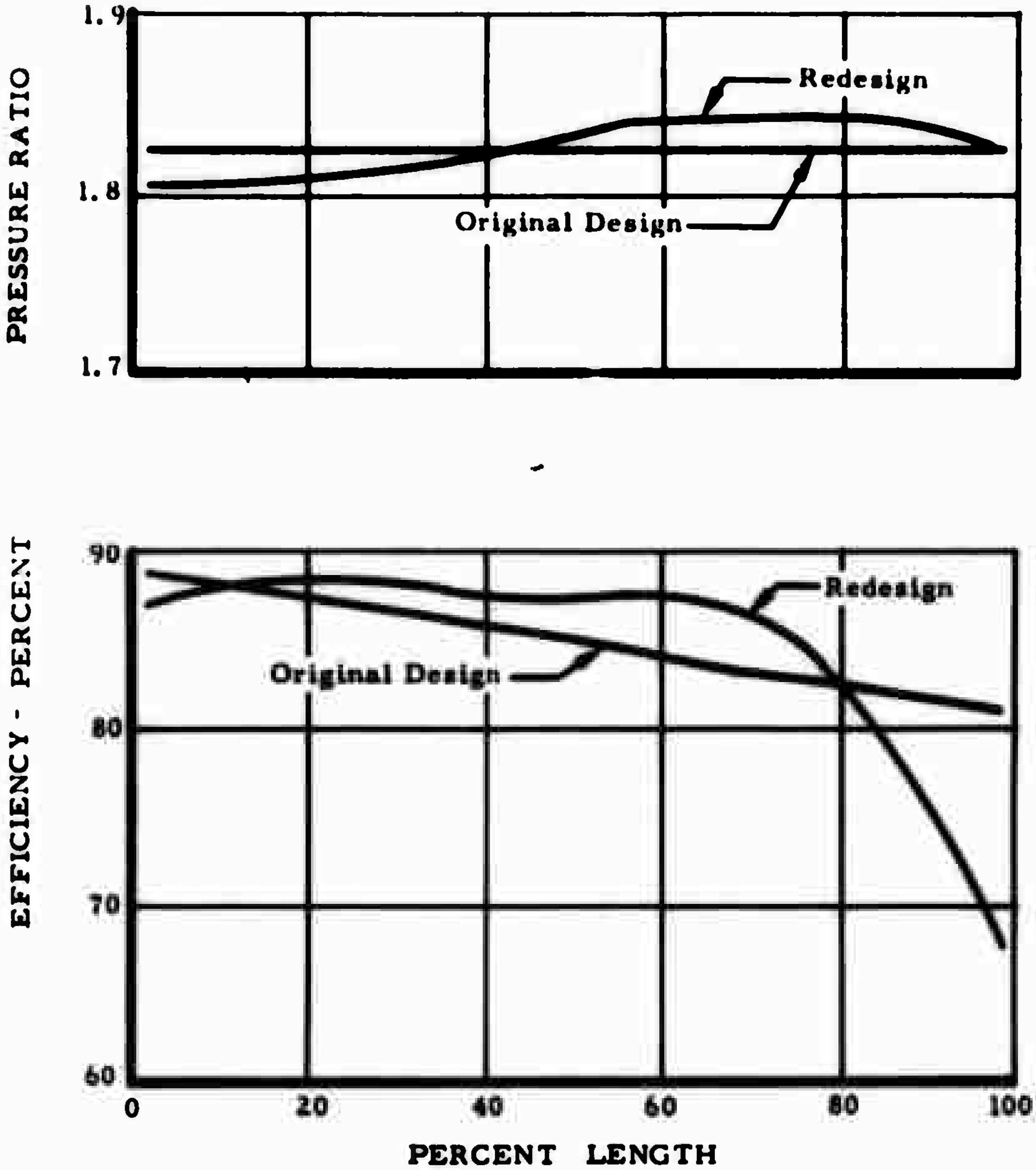

Figure 7. (U) First-Stage Rotor, Performance Comparison.

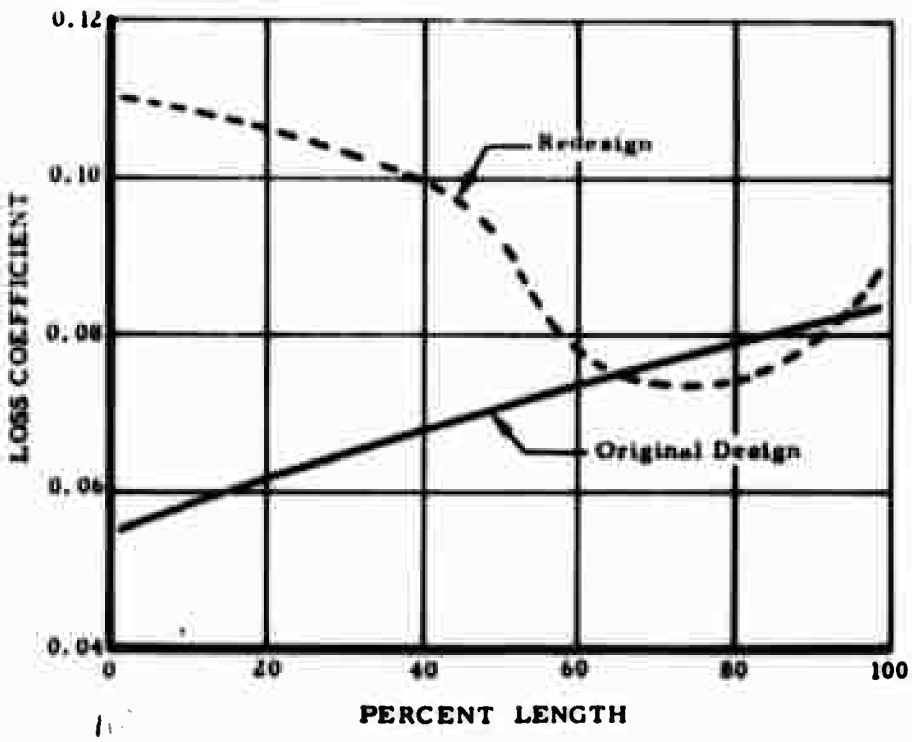

Figure 8. (U) First-Stage Stator, Loss Coefficient Comparison.

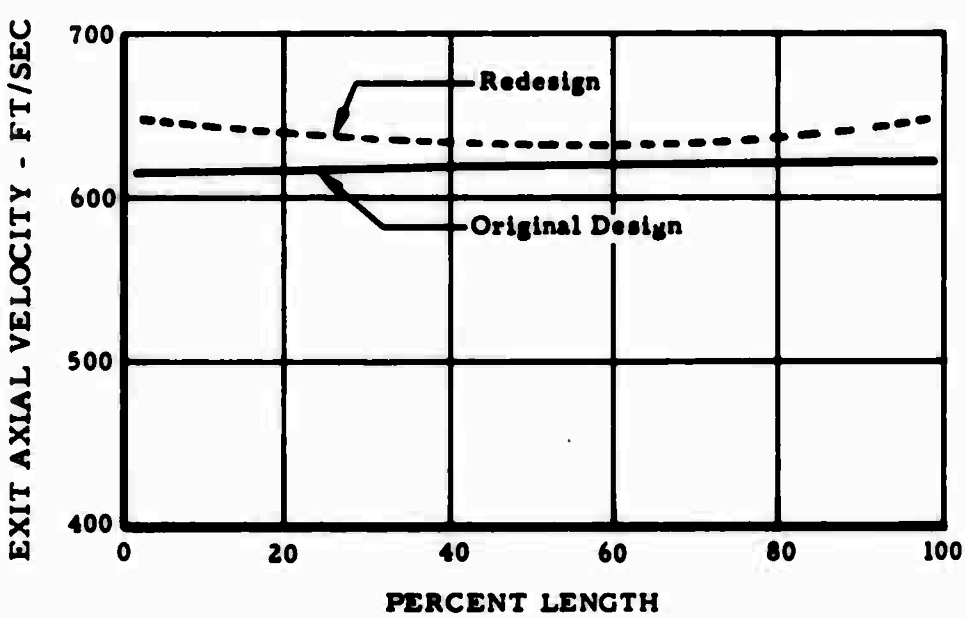

Figure 9. (U) Comparison of Compressor Exit Axial Velocity Profile.

(C) TABLE II (U)

FLOW BLOCKAGE FACTORS (U)

Axial Station	Original Design Blockage	Redesign Blockage
Inlet to Rotor 1	0.99	0.991
Exit of Rotor 1	0.98	0.973
Exit of Stator 1	0.97	0.972
Exit of Rotor 2	0.97	0.954
Exit of Stator 2	0.97	0.953

(U) Good agreement is shown (maximum deviation of 1.7 percent) between the blockage values of the original design and the redesign.

(U) The velocity triangles, which define the aerodynamics of the redesigned first-stage rotor, are presented in Figure 10. The seven velocity triangles are located at selected streamlines that enclose the percent total flow values from the hub as defined below:

Streamline	Total Flow Value From Hub (Percent)
7	100.0
6	83.3
5	66.7
4	50.0
3	33.3
2	16.7
1	0.0

(U) The blade geometry summary is presented in Table III. Double circular arc blading was selected on the basis of flow range and successful experience with other Continental high-pressure-ratio axial compressors.

Streamline 3:

M_1	= 0.580		α_1	=	0.0
M_2	= 0.750		α_2	=	44.70
MX_1	= 0.580		β_1	=	58.94
MX_2	= 0.533		β_2	=	38.51
MR_1	= 1.119				
MR_2	= 0.682				
R_1	= 1.992				
R_2	= 2.124				

Streamline 2:

M_1	= 0.580		α_1	=	0.0
M_2	= 0.784		α_2	=	46.89
MX_1	= 0.580		β_1	=	55.56
MX_2	= 0.535		β_2	=	30.09
MR_1	= 1.017				
MR_2	= 0.619				
R_1	= 1.738				
R_2	= 1.954				

Streamline 1:

M_1	= 0.578		α_1	=	0.0
M_2	= 0.838		α_2	=	50.02
MX_1	= 0.578		β_1	=	50.87
MX_2	= 0.538		β_2	=	16.88
MR_1	= 0.901				
MR_2	= 0.563				
R_1	= 1.438				
R_2	= 1.761				

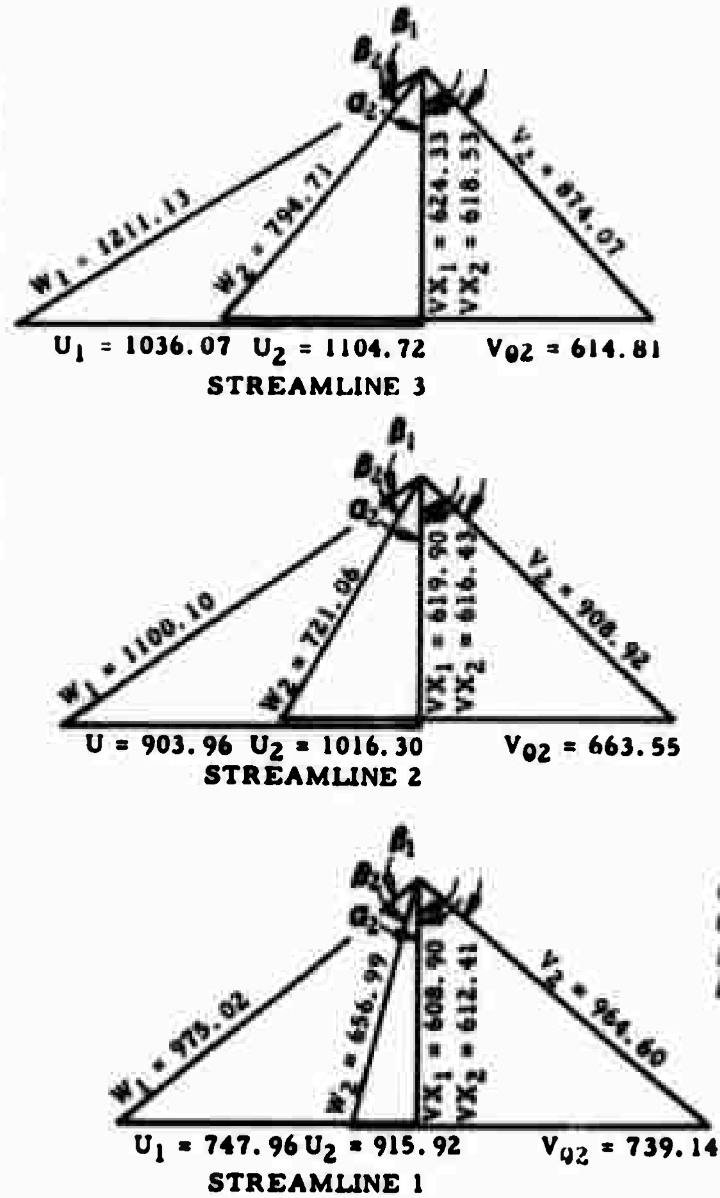

Figure 10. (C) First-Stage Rotor, Velocity Triangles. (U)

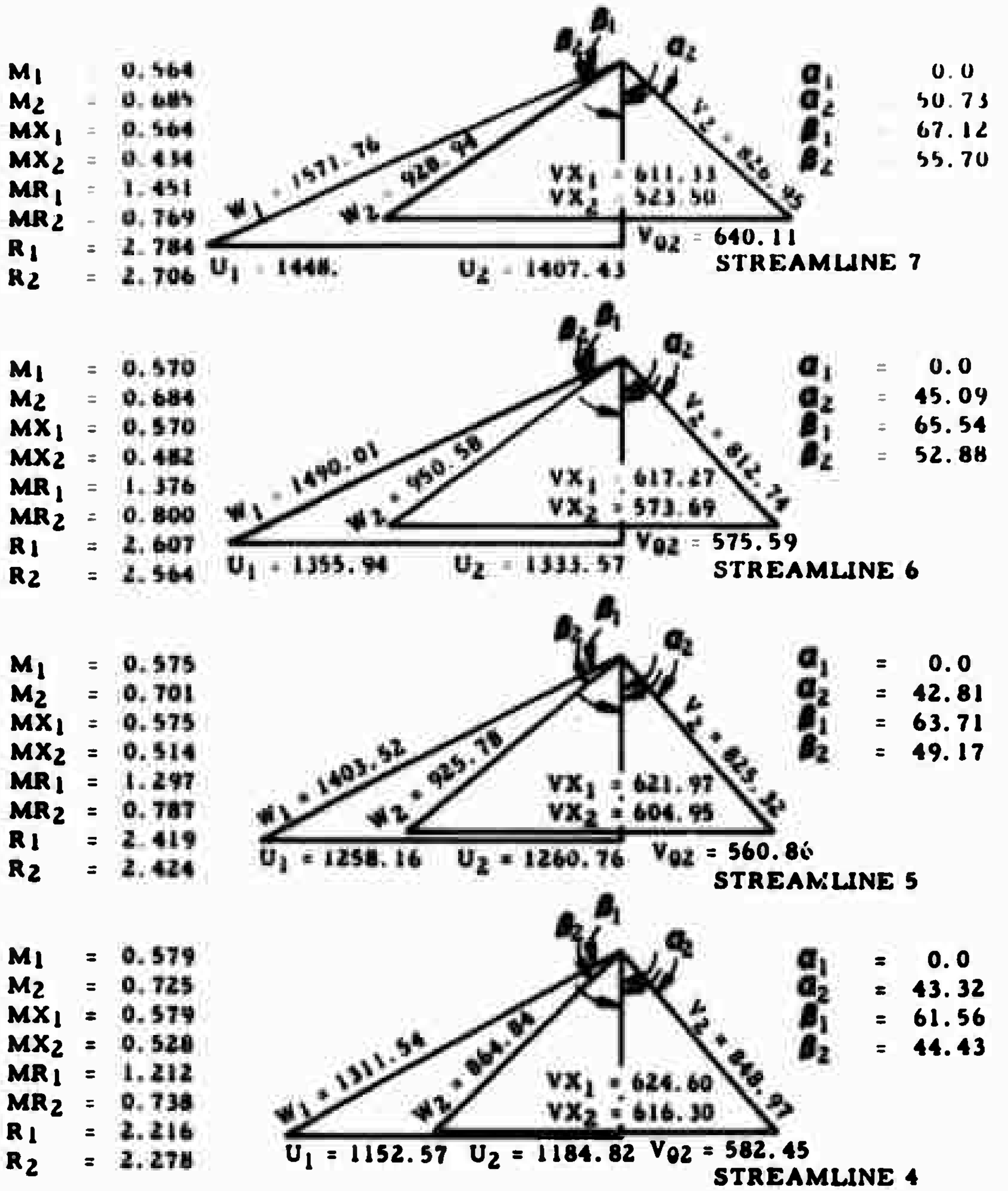

Figure 10. (C) Continued. (U)

(C) TABLE III (U)

USAAVLABS FIRST STAGE ROTOR REDESIGN GEOMETRY, DOUBLE CIRCULAR ARC BLADING (U)

Radius (In.)	Chord (In.)	Solidity	Thickness To Chord Ratio	Suction Surface Radius (In.)	Pressure Surface Radius (In.)	Axial Length (In.)
2.79	1.23	1.403	0.03	6.357	339.71	0.6159
2.50	1.23	1.566	0.0367	3.898	16.89	0.7234
2.20	1.23	1.779	0.0445	2.412	5.85	0.8569
1.90	1.23	2.060	0.054	1.608	3.03	0.9977
1.65	1.23	2.373	0.065	1.161	1.884	1.125

L. E. R. and T. E. R. * To Chord Ratio	Inlet Metal Angle (Deg)	Exit Metal Angle (Deg)	Camber Angle (Deg)
0.00326	62.76	57.14	5.62
0.00326	59.52	48.42	11.09
0.00326	56.18	35.49	20.68
0.00326	52.80	18.77	34.02
0.00326	49.23	-1.594	50.82

Turning Angle (Deg)	Deviation Angle (Deg)	Incidence Angle (Deg)	Setting Angle (Deg)
5.35	3.21	2.94	59.95
9.80	4.67	3.38	53.97
18.01	6.51	3.83	45.83
30.35	7.97	4.30	35.79
47.50	7.99	4.67	23.82

*L.E.R. - Leading edge radius
T.E.R. - Trailing edge radius

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